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SUMMARY

Computations of drag polars for a low-speed Wortmann sailplane airfoil are compared with both wind tunnel and flight test results. Excellent correlation was shown to exist between computations and flight results except when separated flow regimes were encountered. Smoothness of the input coordinates to the PROFILE computer program was found to be essential to obtain accurate comparisons of drag polars or transition location to either the flight or wind tunnel flight results.

INTRODUCTION

The PROFILE computer program used for this study was developed by Dr. Richard Eppler and Dan Somers (ref. 1). PROFILE is used to design and analyze low-speed airfoils. The program is "user friendly" and produces results that correlate closely with results obtained from wind tunnel tests (refs. 1 and 2). These correlations, however, were generally for airfoils that were designed using the PROFILE program. On the other hand, in this report, PROFILE predictions are compared with both wind tunnel and flight results for an airfoil that was not designed using PROFILE.

The airfoil chosen for this study was a first-generation Wortmann, the FX 61-163, which has been used on many sailplanes. Coordinates and previous results from wind tunnel experiments that were used in this report are available in reference 3. Although limited computational analyses were used in the initial design of this 20-year-old airfoil, the final airfoil design was refined primarily using wind tunnel techniques.

This same airfoil, incorporating only slight modifications, was used on a sailplane that was tested in 1979 (ref. 4). The results of that flight test were used for the present study both to independently verify the wind tunnel data and to illustrate the effect that a slight airfoil modification can have on the PROFILE computer program predictions and the flight drag polar.

SYMBOLS

c	test section wing chord, m (ft)
C_d	section drag coefficient
C_ℓ	section lift coefficient
d_1	displacement thickness, nondimensionalized by chord
d_2	momentum thickness, nondimensionalized by chord
d_3	energy thickness, nondimensionalized by chord
H_{32}	boundary layer shape factor, d_3/d_2

R_{d_2} local Reynolds number based on d_2

Re Reynolds number based on test section chord length

q_∞ free-stream dynamic pressure, kN/m^2 (lb/ft^2)

x/c distance along the chord normalized to chord length

y/c distance perpendicular to the chord normalized to chord length

α angle of attack, deg

FLIGHT TESTS

The flight vehicle, described in reference 4 and shown in figure 1, was a T-6 sailplane. The T-6 was an HP-14 design that had been altered to include a T-tail and new wings that used a modified Wortmann FX 61-163 section, referred to in this paper as the flight airfoil. The flight airfoil coordinates from the wing test section are given in table 1. The wing section that was studied was located near mid-span on the right wing (as shown in fig. 1). Modifications included both a plain flap hinged at 79.4-percent chord and a straight-lined lower surface trailing edge. Figure 2 is a comparison of the FX 61-163 airfoil and the flight airfoil showing these modifications.

Flight procedures for taking accurate data necessitated smooth, steady flight at a constant airspeed. Flight at low airspeeds corresponded to a flight condition with a low Reynolds number and high lift coefficient as all data were taken during straight-and-level, 1g trimmed flight. Wing test section Reynolds numbers varied from about 1×10^6 to 3×10^6 as airspeed varied from near 40 knots to 125 knots.

Wake surveys of the wing test section were obtained by installing small pitot and static probes that traversed the wake at a distance of 30-percent chord behind the trailing edge (fig. 3). Reference total pressure was obtained from a kiel tube mounted on the upper wing surface; reference static pressure was obtained from a trailing static located 200-percent chord aft of, and in plane with, the trailing edge. To illustrate the repeatability and accuracy of the data from the wing wake surveys, a typical wake sample from reference 4 is shown in figure 4. The repeatability and low magnitude of the random errors illustrate the low random scatter of this method. Section drag coefficients were computed from the wake data using the method developed by Jones which integrates the momentum deficit in the wing wake (ref. 5). Section lift coefficients were estimated by using the VORTEX-LATTICE program (ref. 6). Estimated total aircraft lift (±1 percent) was adjusted for measured tail loads, and span load was adjusted by measuring flap and aileron deflections.

DESCRIPTION OF PROFILE

Background information is necessary to understand the PROFILE program's ability to predict airfoil section lift and drag polars. Conceptually, the computational analysis is divided into two parts — the inviscid and the viscous. The inviscid part determines pressure coefficient distribution or velocity distribution over the airfoil. The program accomplishes this by using a vortex panel method with parabolically distributed singularities on cubic spline-fitted surfaces between coordinate points. An example of a calculated pressure distribution for the flight airfoil is given in figure 5.

The viscous analysis part of the program uses specified values for Reynolds number and surface roughness to compute transition and separation characteristics of the wing section being analyzed. All airfoils used in this study were considered to have a smooth surface. The viscous part of the program also computes the boundary layer development consisting of the displacement, momentum, and energy thickness (d_1, d_2, and d_3, respectively), as well as the boundary layer shape factor (H_{32}) that can also be obtained. Transition location is also computed from H_{32}.

The location of the transition point from laminar to turbulent flow is considered to be a function only of H_{32}. H_{32} is computed as a function of arc length from the trailing edge. The criteria for transition were empirically developed by Eppler in reference 7 and is given in equation (1).

$$\ln(Rd_2) = 18.4(H_{32}) - 21.78 \tag{1}$$

The correlation between wind tunnel transition location in reference 8 and PROFILE computational values are excellent for the FX 66-AII-182 airfoil as shown in figure 6. The error shown is less than 1-percent chord.

Criteria for flow separation are not as well defined as for transition. Basically, when the value of H_{32} becomes less than 1.46, turbulent separation is assumed. A more involved development of this separation criteria has been detailed by Schlichting (ref. 9). The PROFILE program usually provides a reasonable definition of the separated flow regions. However, once separation exists, it degrades the prediction of section drag.

COMPARISONS

Input coordinate smoothness was found to be critical to the full development of the computed drag bucket. In computation, the raw data T-6 airfoil (ref. 4) used in flight produced the narrowest drag bucket, with the smoothed airfoil producing the widest, as shown in figure 7. Both smoothed and unsmoothed airfoils exhibited large increases in C_d exiting the drag bucket, initially as transition moved to the leading edge (C_ℓ of 0.1 and 0.7 for the raw data T-6 airfoil and -0.1 and 1.0 for the smoothed flight airfoil) and finally as separation began at the trailing edge. Note that little effect is seen in C_d within the drag bucket of the raw data T-6 airfoil when compared with the smoothed airfoil. The smoothed airfoil was within ± 0.00005 y/c of ideally smooth, and it is possible that continued smoothing would further improve the correlation of flight to computed polars. The final smoothing process was accomplished by hand because no computational method has been found that will adequately remove waviness for use with PROFILE. However, when an airfoil is smoothed in this way, the airfoil may not perform as designed.

It appears that the smoothness required by the PROFILE program exceeds that needed for high performance in flight since the raw data T-6 airfoil used the exact wing geometry, yet the T-6 airfoil performance predicted by PROFILE was substantially less than that previously measured in flight.

The PROFILE program was also used to analyze the FX 61-163, called the baseline airfoil, and the smoothed T-6 sailplane airfoil, called the flight airfoil, with and without 6° of flap. Analyses were conducted at Reynolds numbers of 1×10^6, 1.3×10^6, 2×10^6 and 3×10^6.

Figure 8 shows a comparison of the flight airfoil 0° flap deflection results with the baseline wind tunnel results. Note the flight airfoil C_d is less than the baseline airfoil's C_d below 1.0 C_ℓ. The lower surface trailing edge modifications made to the flight airfoil resulted in lower C_ℓ for a given C_d at higher lift coefficients. This reduction was because of the reduced camber of the flight airfoil, resulting in a lower C_ℓ for a given α. With the flight airfoil flap deflected 6°, the flight and baseline airfoils have similar lift coefficients at the same angle of attack because of approximately similar camber lines.

In figure 9, flight results with 6° of flap deflection show improved correlation with the baseline results at high lift coefficients.

In figure 10, the polar results of the baseline airfoil computations show a close correlation to the baseline airfoil tested in the wind tunnel at 2×10^6 and 3×10^6 Re over most of the C_ℓ range. This agreement is good considering that polar accuracies between wind tunnel tests are rarely stated to be better than 2 to 3 percent. Computations of the baseline airfoil for 1×10^6 Re show close correlation to the results of wind tunnel tests only near C_ℓ of 0.25 and 1.05. At intermediate C_ℓ (shown on fig. 7) of 1×10^6 Re the sinuous characteristic of the wind tunnel polar indicates the probable presence of a laminar separation bubble — an effect that would not be accurately predicted using the PROFILE program. Modern airfoils are designed to avoid separation bubbles and, hence, would avoid both the characteristic curvature and inaccurate prediction. Regardless of the cause, the computations do not correlate well with the 1×10^6 Re wind tunnel data. It should be emphasized, however, that for many other airfoils, excellent correlation has been obtained with wind tunnel results as demonstrated in reference 2. Airfoils that have been refined with a "file-it and try-it" approach or modified to simplify fabrication may result in their performance being conservatively predicted. However, they often do not lend themselves to easy analysis because of residual surface waviness from fabrication. This difficulty exists with the flight airfoil.

The flight data with 0° flap is compared to the PROFILE program predictions for the flight airfoil in figure 11. The correlation between flight and analytic results are excellent at Reynolds numbers of 3×10^6 and 2×10^6 up to about 0.7 C_ℓ. As the C_ℓ continues to rise and the Re decreases, the correlation between the flight and analytic results decreases. The large predicted increase in C_d is caused by the beginning of the trailing edge separation.

The flight data with 6° of flap compared with the PROFILE prediction of the flight airfoil with a 6° flap model is shown in figure 12. Again, at low C_ℓ (approximately 0.3) the correlation between flight and analytic results is excellent. However, in this case the agreement remains excellent as low as a Re of 1.3×10^6

and as high as 0.8 C_ℓ. Although the correlation of the analytic results to flight results again decreases at the high C_ℓ, the deviation is not as large as that which occurred in the unflapped case shown in figure 8. Agreement decreases as the trailing edge is predicted to have separation growth.

The above results show that flight performance exceeds the program predictions by a small margin. Even with minor fabrication irregularities in the wing, it appears that predicted performance can be met using current production methods. This predicted performance, however, does not include items that disturb the flow, such as exposed rivet heads or sheet metal lap joints.

GENERAL REMARKS

1. The polar and transition predictions resulting from the PROFILE program provided a high correlation with results of both wind tunnel and flight experiments where separation was not predicted or expected to exist. Furthermore, the performance predicted by PROFILE for an airfoil can be expected to be achieved using current aircraft construction techniques.

2. An operational, user-friendly, computational technique is needed that will provide more accurate predictions of airfoil lift and drag with separated flow.

3. The PROFILE program requires reduced sensitivity to surface waviness, or a smoothing routine to reduce waviness should be incorporated to lessen or eliminate the need for hand smoothing of airfoil coordinates.

4. Flight techniques need to be improved. In the past, flight techniques that have been used to measure airfoil characteristics include wake rakes, hot films, subliming chemicals, and pressure orifices. Although each of these techniques has valid uses, they are of limited application. The serial use of these, plus a few other techniques, are required to measure a full set of airfoil characteristics.

CONCLUDING REMARKS

Drag polars of the airfoil section were computed using the PROFILE program for an FX 61-163 baseline airfoil and for a similar airfoil flown on a sailplane. Compared with the results obtained from wind tunnel data on the baseline airfoil, the computations indicated close correlation with results from wind tunnel experiments at moderate and high Reynolds numbers and poor correlation with the results from wind tunnel experiments at the 1×10^6 Reynolds number where a probable separation anomaly existed. Compared with the flight data, the computations were very good for both the flapped and unflapped cases at low and moderate section lift coefficients but were not as accurate at higher lift coefficients where PROFILE predicted the beginning of separated flow at the trailing edge. Poor correlation was shown to exist when the airfoil being analyzed was insufficiently smooth. On the other hand, greatly improved correlation resulted from extensive smoothing of the computed flight test airfoil. A certain degree of smoothness is necessary because of PROFILE's sensitivity to surface waviness, but is not needed for good performance on flight airfoils. The performance predicted by PROFILE is easily achieved with existing fabrication technologies.

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Edwards, California, July 27, 1984

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TABLE 1. — COORDINATES OF THE SMOOTHED FLIGHT AIRFOIL WITH 0° FLAP DEFLECTION

x/c (upper)	y/c	x/c (lower)	y/c
0	0.00243	0	-0.00243
0.00224	0.00882	0.00087	-0.00320
0.00734	0.01595	0.00519	-0.00806
0.01502	0.02355	0.01327	-0.01283
0.02529	0.03156	0.02465	-0.01775
0.03787	0.03975	0.03896	-0.02261
0.05265	0.04800	0.05619	-0.02750
0.06951	0.05602	0.07567	-0.03210
0.08851	0.06382	0.09807	-0.03651
0.10964	0.07133	0.12272	-0.04076
0.13217	0.07808	0.14958	-0.04489
0.15663	0.08420	0.17836	-0.04894
0.18308	0.08970	0.20818	-0.05252
0.21115	0.09453	0.23979	-0.05540
0.24027	0.09856	0.27263	-0.05768
0.27058	0.10164	0.30685	-0.05953
0.33933	0.10506	0.33933	-0.06072
0.37056	0.10495	0.37056	-0.06124
0.40243	0.10388	0.40243	-0.06100
0.43469	0.10188	0.43469	-0.05871
0.46733	0.09888	0.46733	-0.05669
0.49997	0.09515	0.49997	-0.05428
0.53274	0.09045	0.53274	-0.05138
0.56525	0.08516	0.56525	-0.04819
0.59750	0.07939	0.59750	-0.04474
0.62938	0.07311	0.62938	-0.04109
0.66074	0.06692	0.66074	-0.03749
0.69133	0.06075	0.69133	-0.03398
0.72115	0.05472	0.72115	-0.03059
0.74995	0.04884	0.74995	-0.02732
0.77773	0.04328	0.77773	-0.02419
0.80435	0.03791	0.80435	-0.02121
0.82970	0.03279	0.82970	-0.01841
0.85350	0.02806	0.85350	-0.01577
0.87590	0.02366	0.87590	-0.01333
0.89664	0.01963	0.89664	-0.01109
0.91571	0.01600	0.91571	-0.00905
0.93299	0.01284	0.93299	-0.00723
0.94848	0.01005	0.94848	-0.00565
0.96192	0.00762	0.96192	-0.00318
0.98291	0.00383	0.98291	-0.00117
1.00000	0.00083	1.00000	-0.00117

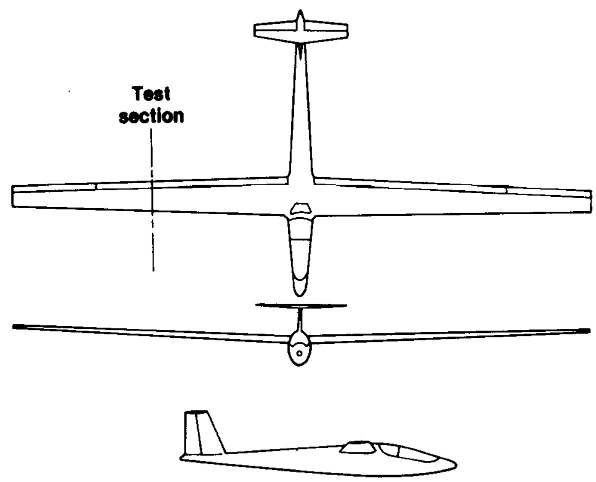

Figure 1. Three view of T-6 sailplane and test section location. Span = 14.93 m (49 ft); weight = 367 kg (830 lb); area = 13.2 m^2 (129 ft^2); and test section chord = 75.9 cm (29.875 in).

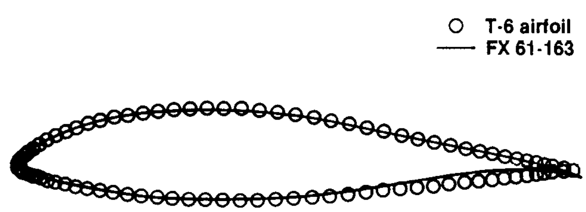

Figure 2. A comparison of the baseline FX 61-163 airfoil and the T-6 flight airfoil with 0° flap deflection.

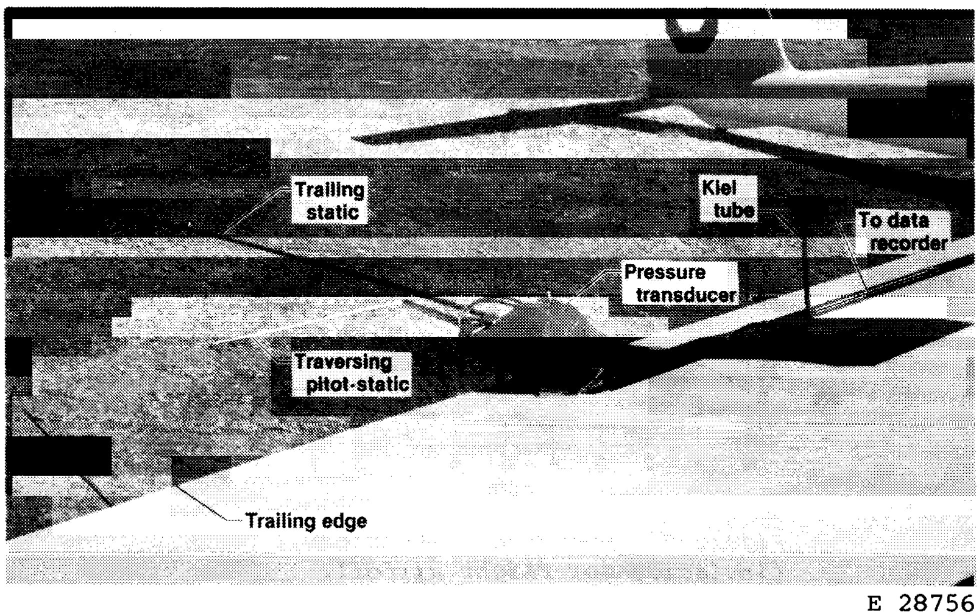

Figure 3. Wake rake installation on T-6 sailplane.

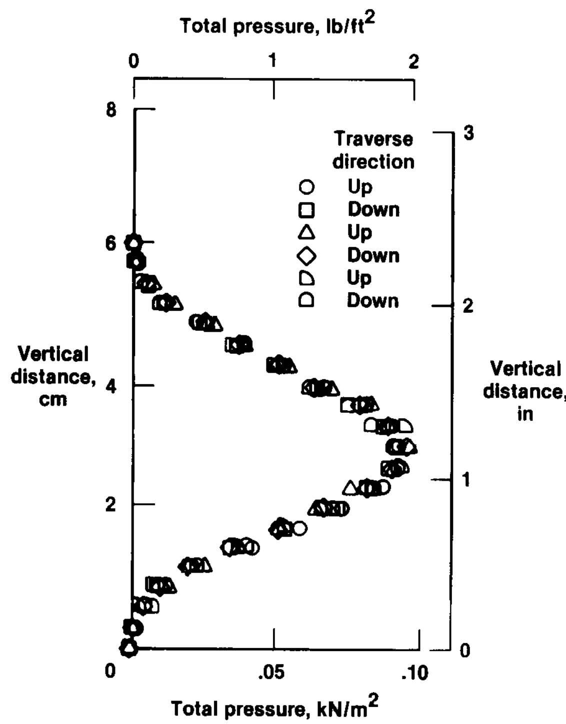

Figure 4. Typical total pressure wake profiles. Six consecutive wakes; flap deflection = 0°; velocity = 44.0 knots; q_0 = 0.31 kN/m^2 (6.5 lb/ft^2). (From ref. 4.)

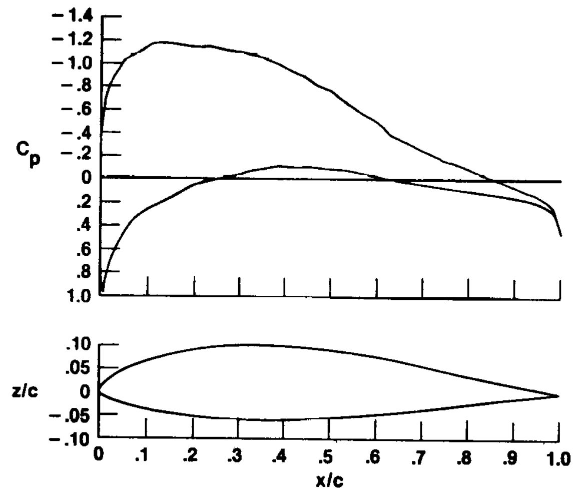

Figure 5. Pressure coefficient (inviscid) for flight airfoil, $\alpha = 4°$.

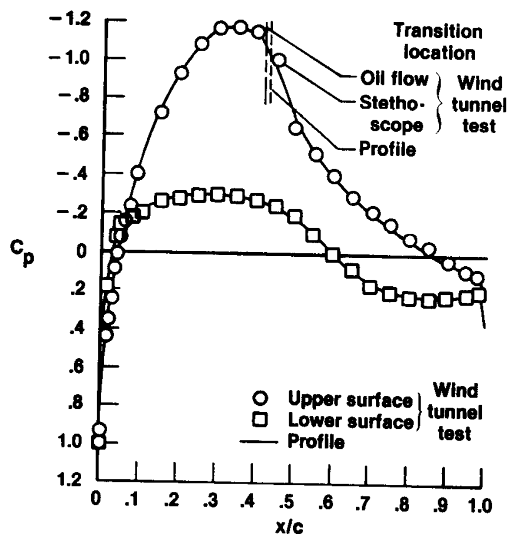

Figure 6. Transition location and pressure coefficient comparison for FX66-4II-182 airfoil in the wind tunnel at Re = 1.5×10^6, $\alpha = 0°$, and $C_\ell \approx 0.4$; low-speed airfoil program at Re = 1.5×10^6, $\alpha = 0°$, and $C_\ell = 0.368$.

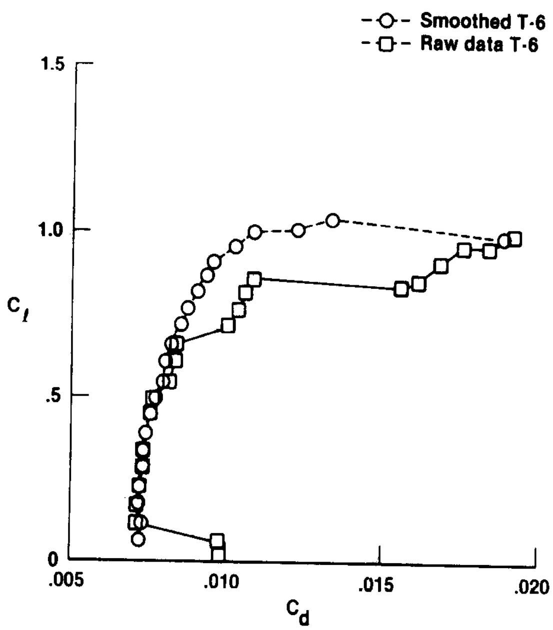

Figure 7. A comparison of computed polars of the raw data T-6 flight airfoil and the final smoothed T-6 airfoil. Re = 1 × 10^6 and 0° flap deflection.

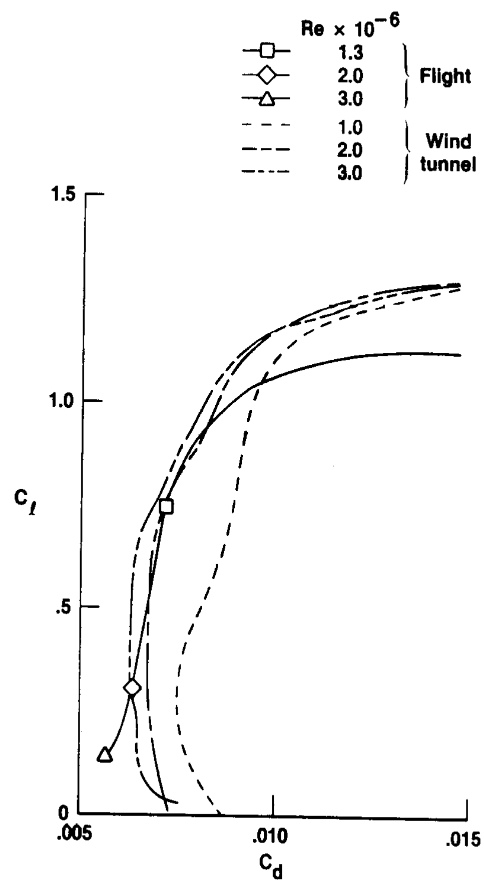

Figure 8. A comparison of flight data (ref. 4) with 0° flap deflection and the baseline wind tunnel data (ref. 3).

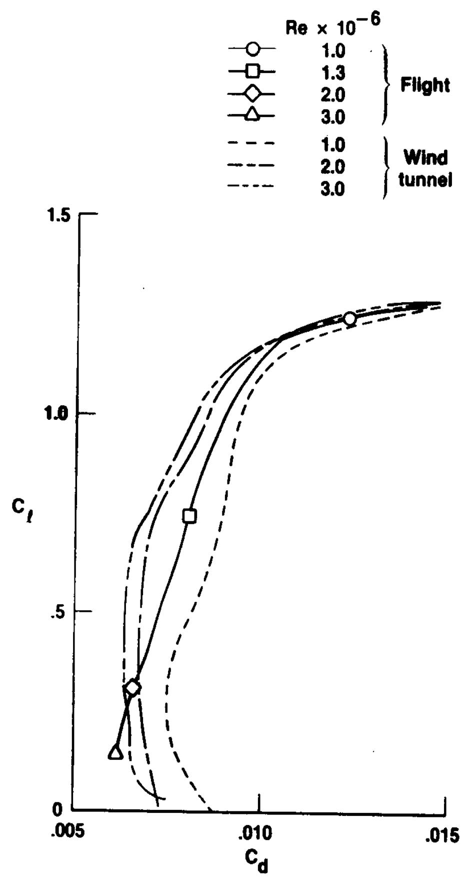

Figure 9. A comparison of flight data (ref. 4) with 6° flap deflection and the baseline wind tunnel data (ref. 3).

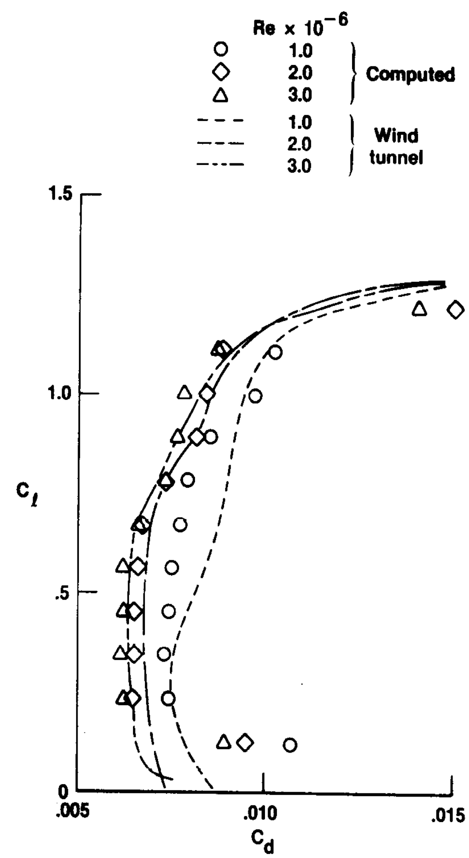

Figure 10. A comparison of computed and wind tunnel data (ref. 3) for the baseline airfoil.

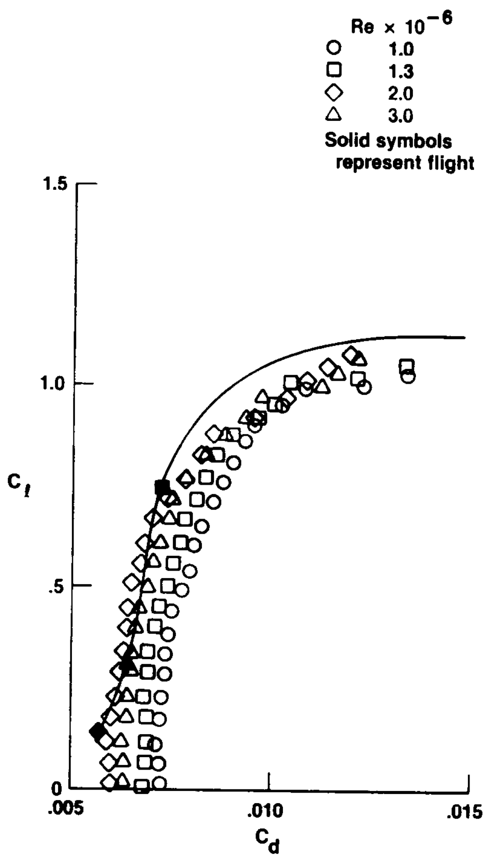

Figure 11. A comparison of computed and flight data (ref. 4) for the flight airfoil with 0° flap deflection.

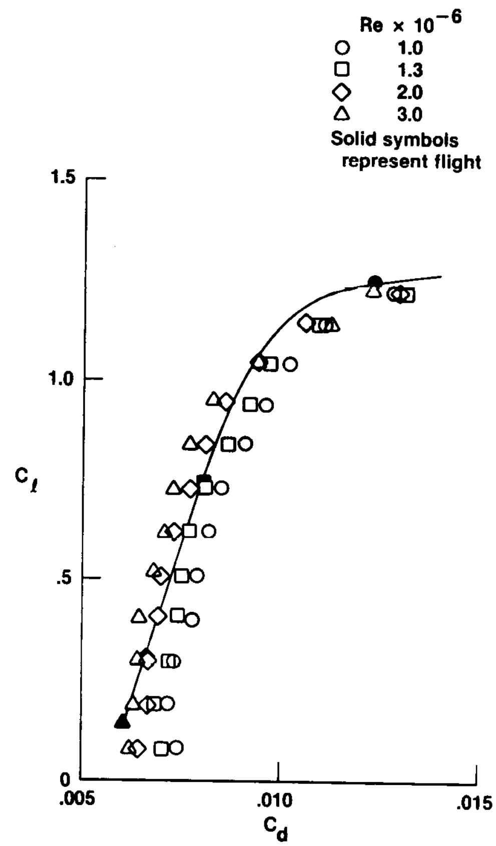

Figure 12. A comparison of computed and flight data (ref. 4) for the flight airfoil with 6° flap deflection.

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16. Abstract

Computations of drag polars for a low-speed Wortmann sailplane airfoil are compared with both wind tunnel and flight test results. Excellent correlation was shown to exist between computations and flight results except when separated flow regimes were encountered. Smoothness of the input coordinates to the PROFILE computer program was found to be essential to obtain accurate comparisons of drag polars or transition location to either the flight or wind tunnel flight results.

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